AD NUMBER
AD007922
CLASSIFICATION CHANGES
TO: unclassified
FROM: restricted
LIMITATION CHANGES
TO: Approved for public release, distribution unlimited
FROM: Distribution authorized to U.S. Gov't. agencies and their contractors; Administrative/Operational Use; 24 OCT 1952. Other requests shall be referred to Air Force Rocket Propulsion Laboratory, Edwards AFB, CA.
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AEROPHYSICS DEVELOPMENT CORPORATION

Pacific Palisades, Calif.

Date: 24 October 1952
Report No.: ADC-102-5
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QUARTERLY PROGRESS REPORT

PRELIMINARY PERFORMANCE ANALYSIS

OF THE PULSE-DETONATION-JET ENGINE SYSTEM

Prepared by: D Bitondo
D. Bitondo

Approved by: W. Bollay
W. Bollay

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PRELIMINARY PERFORMANCE ANALYSIS OF THE PULSE-DETONATION-JET ENGINE SYSTEM

24 October 1952

FOREWORD

This report was prepared by the Aerophysics Development Corporation under U.S. Air Force Contract Number AF 33(616)-37. This is the third progress report of the work completed by 10 October 1952 under the research and development contract identified by Expenditure Order No. R-467-4 BR-1. The report is the third of a series to be issued on this project, the first having been submitted on 1 April 1952, the second on 24 July 1952 and the fourth and final progress report due on 24 January 1953. Two technical reports have been submitted besides the above technical reports, one on 26 August 1952 and the other on 18 October 1952.

Included among those who cooperated in these preliminary studies is E. L. Kumm, who assisted with the combustion, heat transfer and fuel control system.

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ABSTRACT

The final performance analysis of the supersonic Pulse-Det-Jet is given in this report. The performance of the engine given in this report represents a revision of the previous results given in Reference 1. These final computations include the new ideas of scavenging flow and minor corrections to the computations. The performance as presented in its final form here does not substantially differ from the results given in the last progress report (Reference 1). The curves of the drag coefficient of a typical supersonic long range missile show, as before, that four 36" diameter engines have enough reserve power to propel the missile through sonic flight velocities and up to a flight Mach number of 2.80.

In addition a unit small enough to be mounted on the blade tip of a helicopter was analysed. This unit has a maximum diameter of 8¼" with combustion tubes 6" long and ranging in diameter from 0.60" to 0.25". The total weight of one unit is approximately 35 pounds. The basic difference of this smaller unit and the large unit previously described is the method of ignition of the fuel. It is doubtful if detonation can be supported in such tubes. But, on the other hand, since the tubes are composed of ceramic materials and are uncooled the combustion is achieved rapidly by surface combustion from the hot walls of the ceramic tubes, the combustion proceeding radially inward in the tube.

This jet unit of 8¼" diameter produces a thrust of 110 lbs at a maximum temperature of 2000°F and has a specific fuel consumption of 1.65 $\frac{\text{lbs}}{\text{HR}}$ / LB. of thrust. Important performance points are given in the following table.

TEMPERATURE °F		STATIC THRUST Pounds	STATIC SPECIFIC FUEL CONSUMPTION Pounds Per Hour Per Pound of Thrust
	1500	75	1.31
Cruising Temperature	1800	93	1.32
	2000	110	1.33
Temperature of Max. Power	2500	135	1.46

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The Multi-Jet engine promises to be a very simple and inexpensive jet unit suitable particularly for helicopters and other subsonic aircraft and missile applications. Preliminary design studies indicate that although the specific fuel consumption of the Multi-Jet is more than that of the reciprocating engine, the much lower weight makes it possible for a Multi-Jet propelled helicopter to carry more payload for long as well as short range operations.

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INTRODUCTION

Two technical reports have been published on the Multi-Jet engine (References 3 and 4). These two reports describe in detail the operation and the performance of the Multi-Jet engine utilizing as ignition source the hot ceramic walls of the tubes with the burning proceeding radially in the tubes.

The performance computations of the supersonic Pulse-Det-Jet utilizing detonation are revised and these are described in detail in this report. The method of computation of these final results is similar to that used in the previous progress report (Reference 1) and the method is reported in full detail here, including tabular data.

The description of the experimental investigation given in the previous report (Reference 1) is amplified and continued in this report

SECTION I

FINAL PERFORMANCE ANALYSIS OF SUPERSONIC ENGINE

1.1 General

The last progress report (Reference 1) gave the complete description and the performance of an engine suitable for supersonic flight. This engine was able to produce (a) static thrust comparable to a turbo jet but its efficiency was not as good and (b) supersonic thrust comparable to a ram jet but at a much better specific fuel consumption.

In computing the performance of the engine as described in Reference 1, various assumptions were made. These were:

(1) Burning time was 0.0015 seconds

(2) A Mach number of 1.0 was assumed for the scavenged gases as they are discharged

(3) The following diffuser efficiencies were used:

Speed	Total Pressure Ratio
$0 \leq M_o \leq 1.0$	1.00
$M_O = 2$	0.95
$M_o = 2.80$	0.65

(4) The thrust was computed from only the impulse obtained (a) from the high pressure sonic discharge of the burnt gases through the open end of the tubes with no nozzle expansion considered, (b) from the discharge of the remaining burnt gases during scavenging.

The impulse of the intake air was subtracted from the above impulses to give the net impulse. The flow during the discharge or expansion phase was assumed to issue at a Mach number of 1.0 and no expansion was considered. Any reaction of the pressures on the solid portions between the tubes was neglected. It is

expected that a pressure greater than static pressure is obtained in the outlet duct of the whole engine.

A study was made of these assumptions to see if they could be improved and what was the effect of any changes.

1.2 Assumptions

It was felt that the burning time used was as good an assumption as can be made at this time. It is hoped that future experimental results in the literature will throw more light on this phase of the cycle.

A better consideration for the scavenge flow is given for the lower flight Mach numbers. For these cases it was felt that the exit velocity for the remainder of the burnt gases was assumed to be too high and therefore gave an impulse that was too great. It will be assumed that this flow is discharged at a velocity equal to the inlet velocity of the fresh fuel-air mixture entering the tube at the front end. This would be true for the lower Mach numbers $0 \leq M_0 < 1.0$.

Consider a tube just at the end of the discharge or expansion phase. The pressure in the tube drops to the total pressure of the inlet flow M_0 at which time the inlet valve is opened. The inlet flow travelling at a Mach number M_0 has been brought to rest at the mouth of the tube while it was closed. Meanwhile, at the exit of the tube, the ambient static pressure is lower than the pressure in the tube, which is equal to total pressure of a flow of Mach number M_0. The conditions here are similar to the subsonic flow surrounding an airfoil, i.e. total pressure at the stagnation point on the leading edge and static pressure at the trailing edge. Therefore, in the tube the remainder of the burnt gases continue to flow out of the tube. This flow may be compared to the steady flow of air in a stream tube towards a stagnation point and then away from the stagnation point.

In computing the impulse produced by the ejection of the remainder of the burnt gases during the scavenging phases the exit velocity is assumed to be the same as the inlet velocity of the new fuel-air mixture. This will be true for the subsonic flight Mach numbers, i.e. $0 \leq M_0 < 1.0$.

For the supersonic Mach numbers it will be seen that the total pressure in the tube will be sufficiently high to produce sonic exit velocity during scavenging. At supersonic flight speeds the pressure in the tube during scavenging and immediately after the discharge phase will be, at first, low, then after the pressure wave from the opening action of the inlet arrives, the pressure is increased. This can be seen in Figures 5 and 6 in Reference 1.

The performance computations are carried out in Appendix I and are tabulated in Tables I and II. The curves for the thrust, specific fuel consumption, specific thrust and thrust per unit maximum frontal area are given in Figures 4, 5, 6, and 7.

The drag coefficient of a typical large missile is plotted on Figure 17(a) in Reference 1. This missile has a ground launch weight of 52,000 pounds with a wing area of 425 square feet. Four engines are required, each having an outside diameter of 36" and each weighing about 1,000 pounds for a total engine weight of 4,000 pounds which is about half that of turbo-jets with after-burner having the same thrust at high supersonic speeds. It will be noted in Figure 8 that this system has sufficient excess thrust to accelerate through $M = 1.0$ at sea level by operating at a maximum cycle temperature of about 3,200°F. The minimum cruising speed at $h = 35{,}000$ feet will be $M_0 = 1.80$ for $T_{MAX} = 2540°F$. Similarly at 5,000 feet, this system has sufficient thrust to accelerate through $M = 1$ and accelerate to $M = 2.80$ by operating at $T_{MAX} = 3350°F$. Figure 9 gives the minimum flight velocity (or take-off velocity).

The maximum cross-sectional diameter of the engine is 36 inches, the length is 55 inches. The cross-sectional area of the combustion tubes is 506 square inches or 51.5% of the maximum cross-sectional area of the engine. There are six concentric rows of 32 tubes each. The diameters of the tubes decrease from the outer to the inner circle and they are 2.62", 2.22", 1.87", 1.50", 1.23", and 1.02". The length of the tubes are 15". The total weight of the engine is approximately 950 lbs.

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FIGURE 4

ACTUAL THRUST OF A 36" DIAMETER PULSE-DET-JET

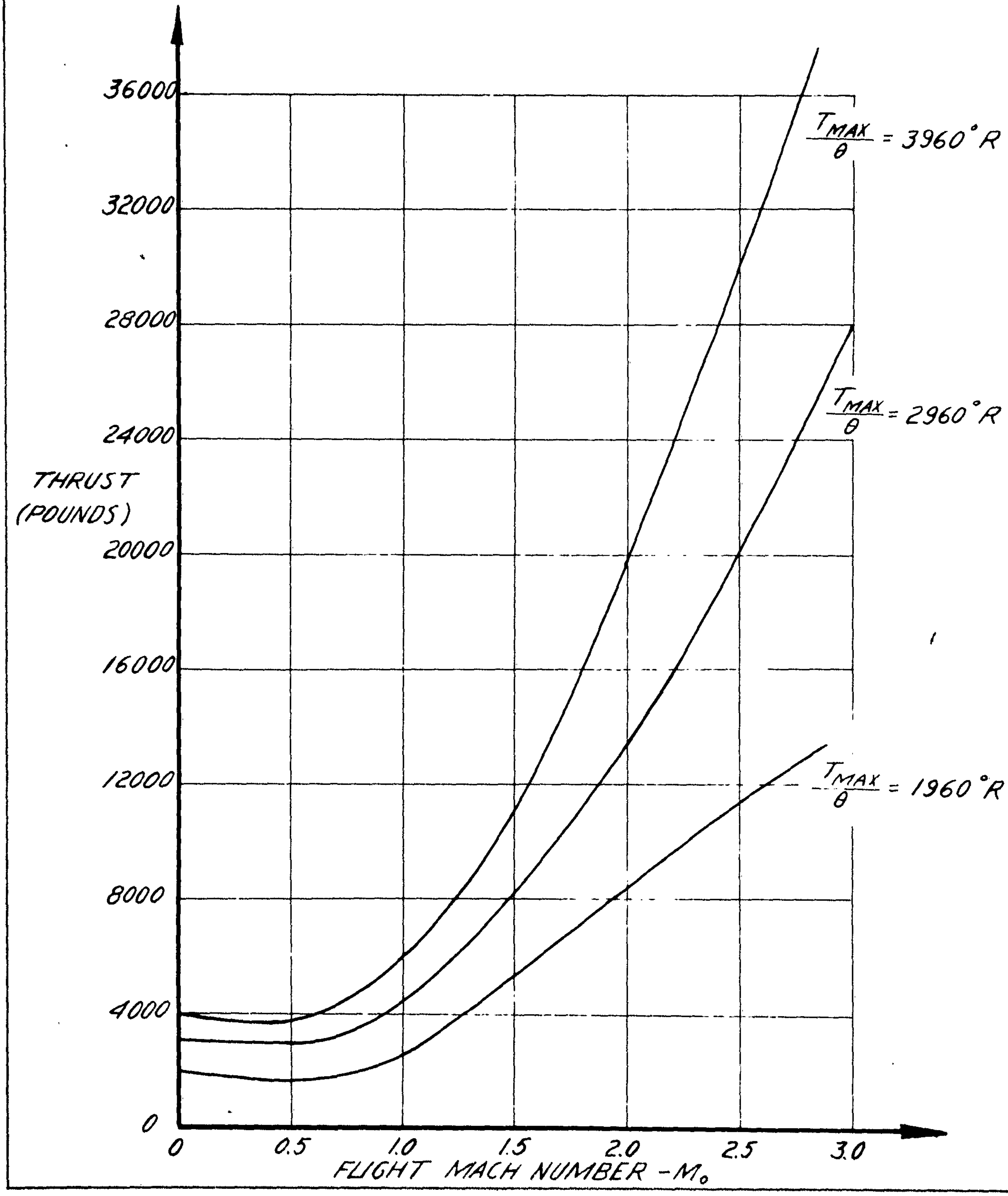

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FIGURE 5

ACTUAL SPECIFIC FUEL CONSUMPTION OF THE PULSE-DET-JET

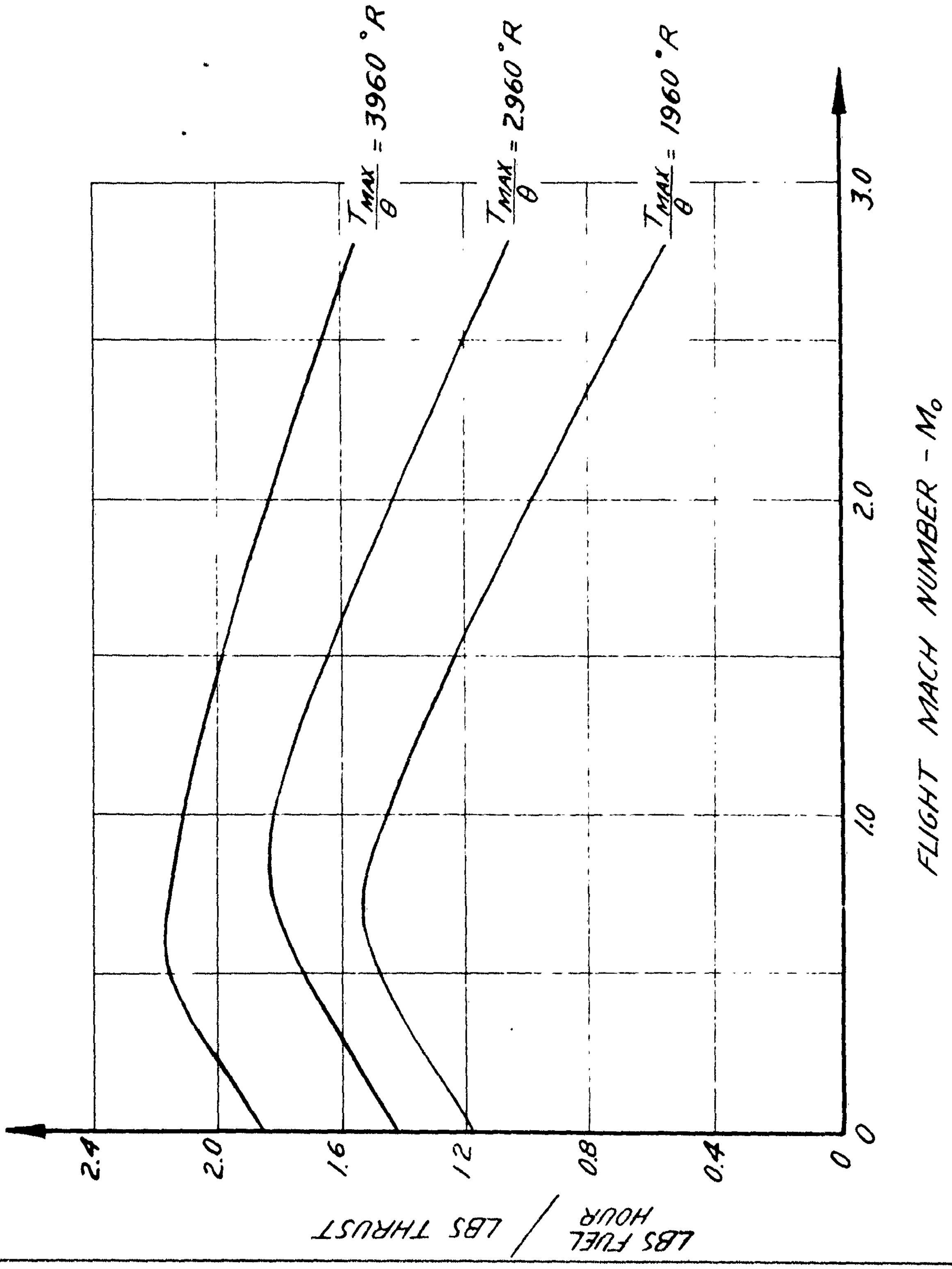

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FIGURE 6
ACTUAL SPECIFIC THRUST OF THE PULSE-DET-JET

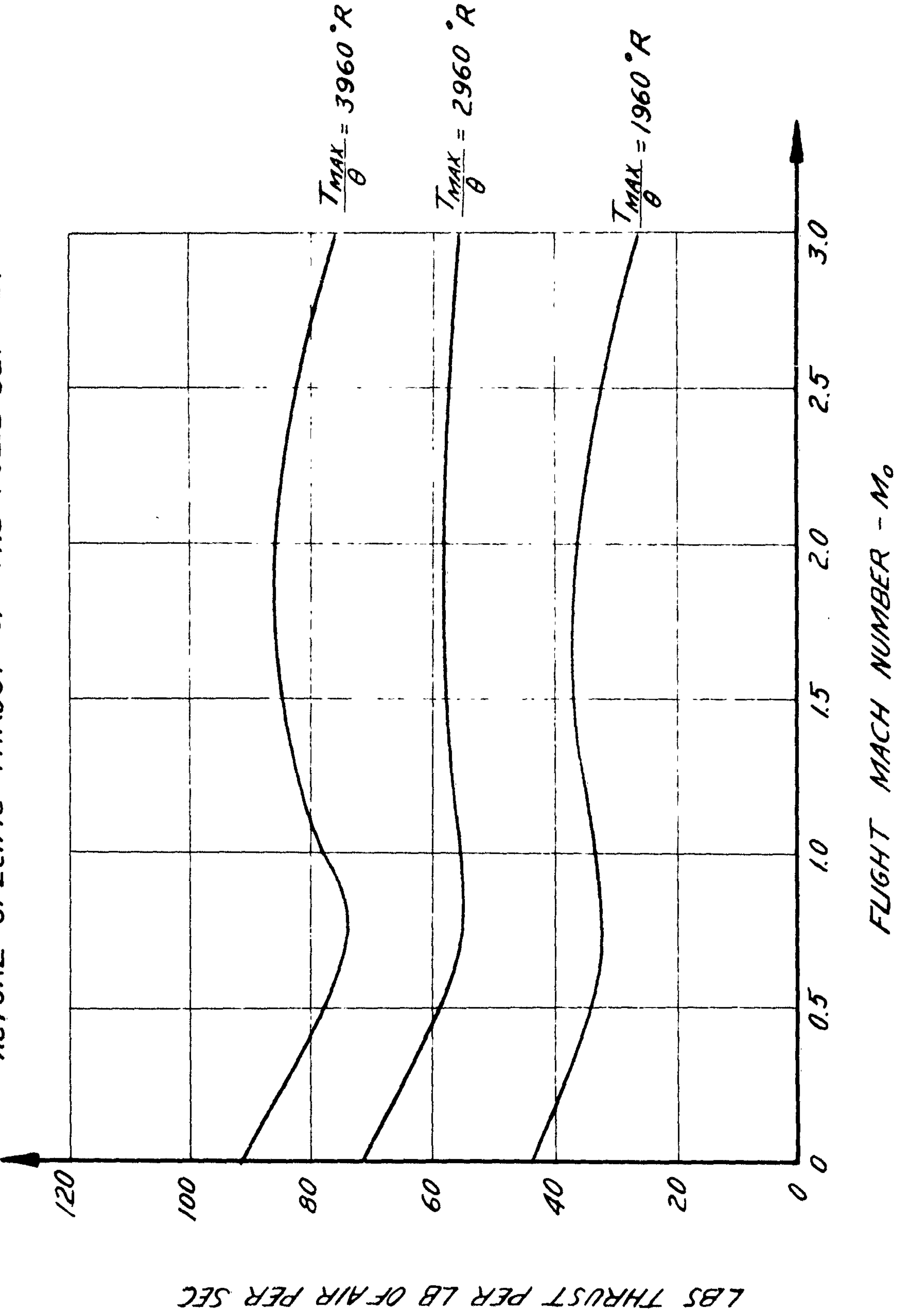

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FIGURE 7

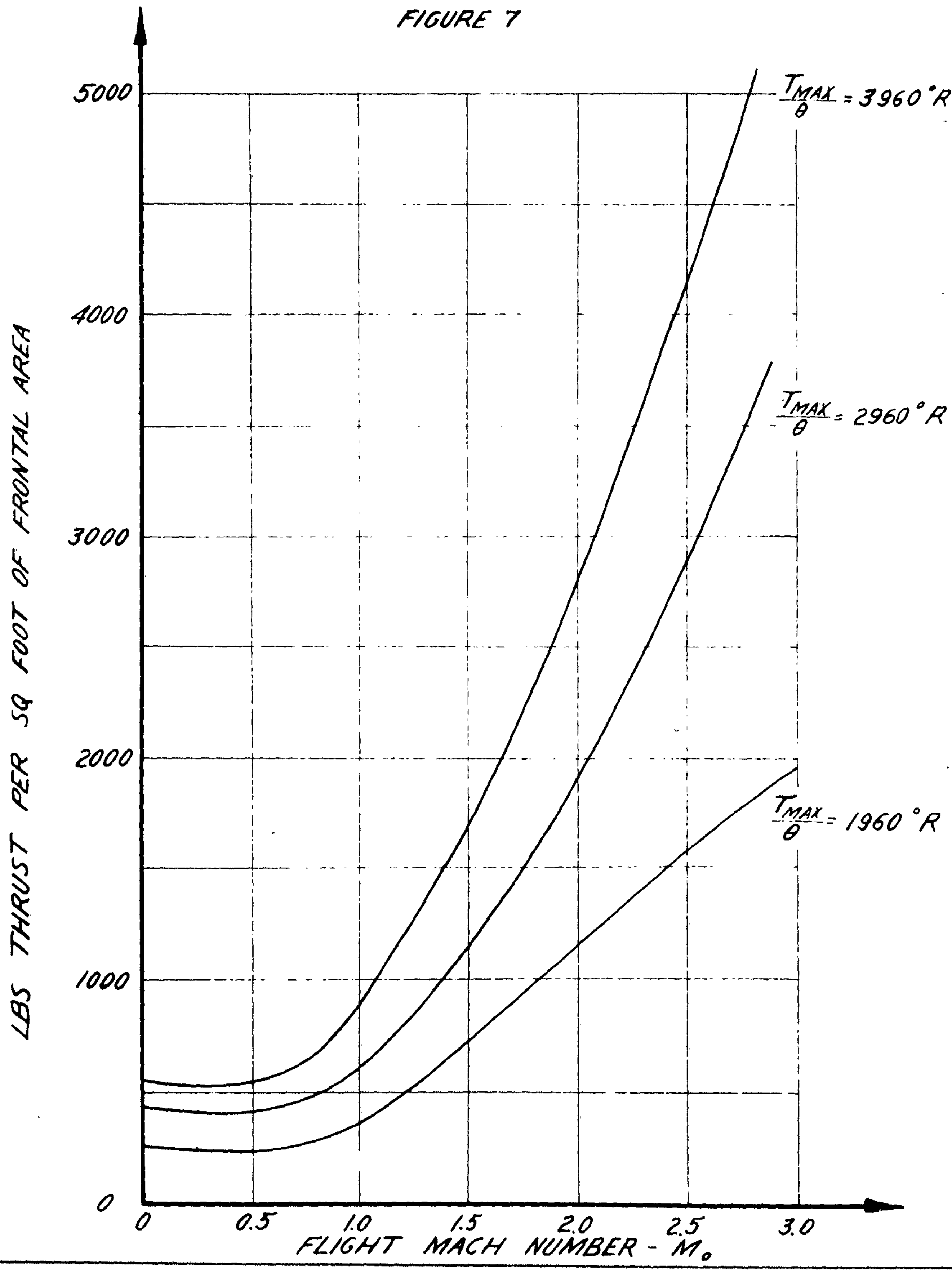
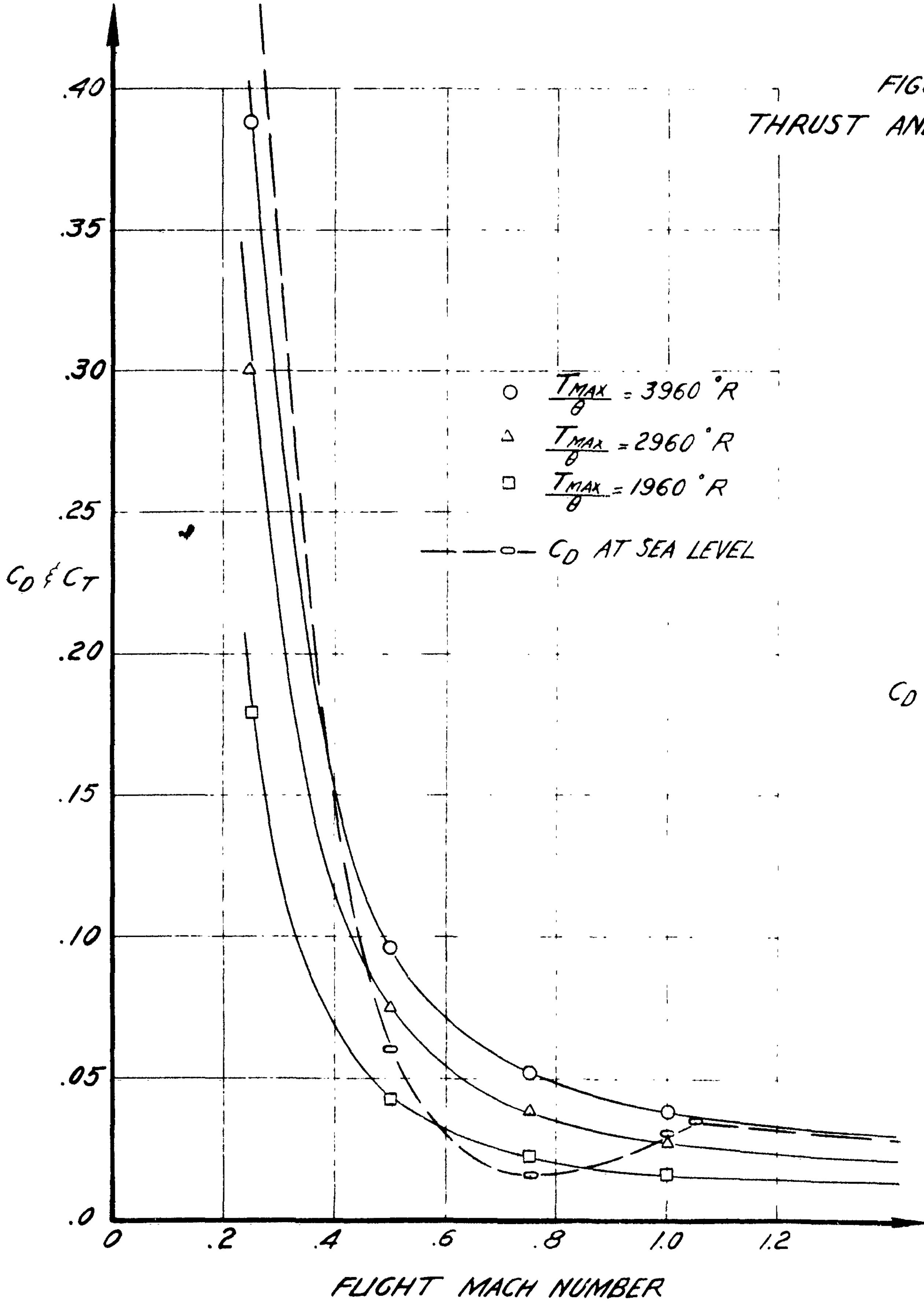
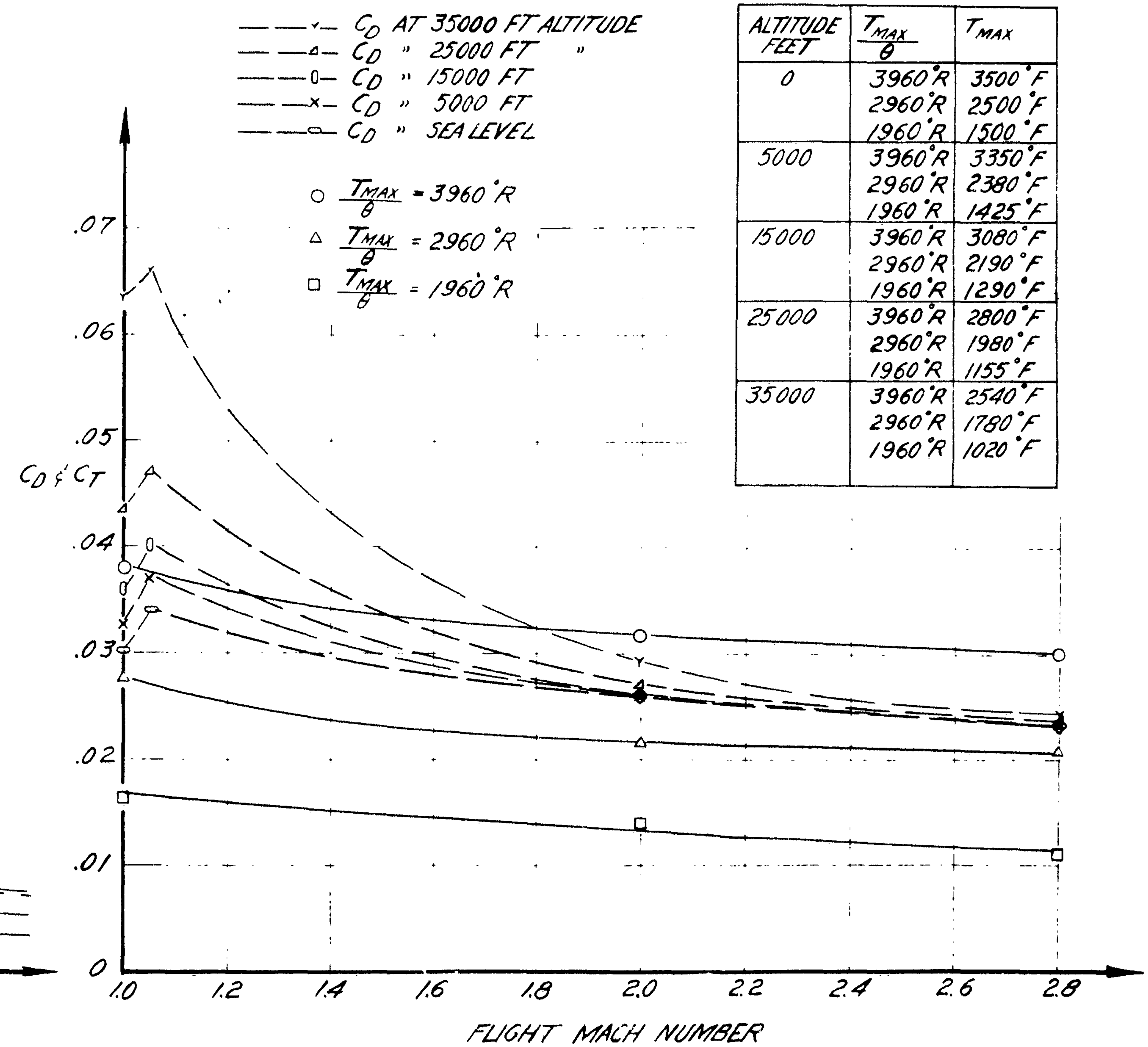


FIGURE 8
THRUST AND DRAG COEFFICIENTS





ALTITUDE FEET	$\frac{T_{MAX}}{\theta}$	T_{MAX}
0	3960°R	3500°F
	2960°R	2500°F
	1960°R	1500°F
5000	3960°R	3350°F
	2960°R	2380°F
	1960°R	1425°F
15000	3960°R	3080°F
	2960°R	2190°F
	1960°R	1290°F
25000	3960°R	2800°F
	2960°R	1980°F
	1960°R	1155°F
35000	3960°R	2540°F
	2960°R	1780°F
	1960°R	1020°F

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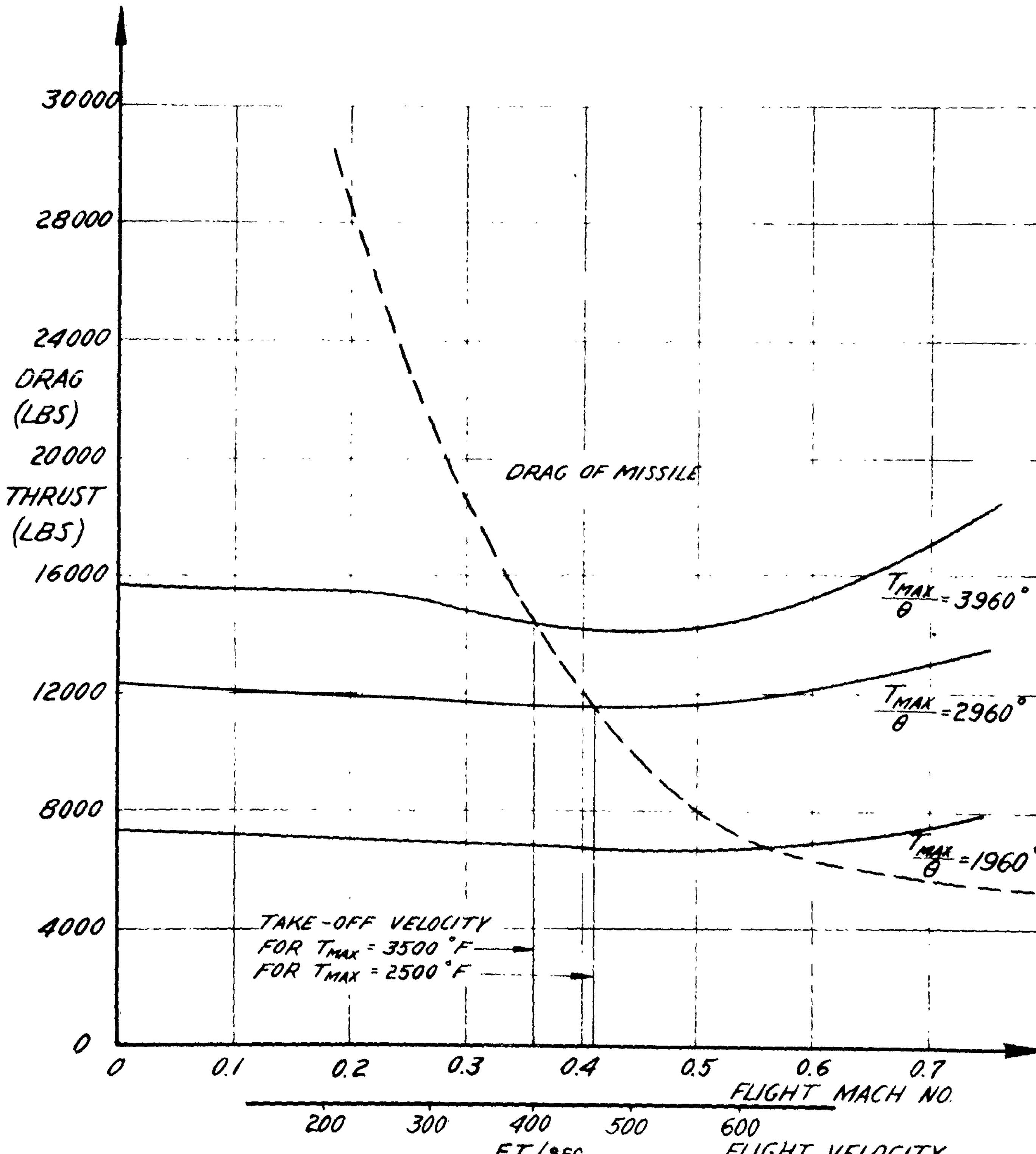
FIGURE 9

DRAG AND THRUST AT SEA LEVEL



SECTION II

ANALYSIS OF THE SUBSONIC ENGINE

2.1 Introduction

Two technical reports have been written on the preliminary analysis of an engine suitable for helicopter rotor propulsion. A typical engine is described having an external diameter of 8½ inches, an overall length of about 16 inches with 50% of the maximum area as combustion tube area. The tubes are made of a ceramic material suitable to withstand the internal pressures at high wall temperatures.

The basic difference between this engine and the supersonic engine is the method of ignition of the fuel. Since the tubes are of such a small diameter and length, it is doubtful if detonation can be initiated or even supported under such conditions. On the other hand, by taking advantage of the ignition and flameholding qualities of the hot walls of a ceramic tube this difficulty can be overcome. References 5 and 6 describe a method of burning at very high rates of mass flow within ceramic burners. In reference 5 at the University of Michigan very high values of heat release per cubic foot of combustion chamber space was obtained by burning a stream of fuel mixture travelling at very large velocities through the burner. In Reference 6 at the Gas Research Board very high values of heat release per cubic foot of combustion chamber space were also obtained. The flow velocity in Reference 6 is of the same magnitude as that in present ram jet burners, but much shorter combustion chamber lengths were used. In Reference 5 the main improvement of the ceramic burner over the present normal burners is due to the very high velocity employed in the burner tests.

It is proposed to make use of this phenomenon of surface combustion. If it turns out that the mechanism of burning is obtained through a series of explosions then by careful design of the tube lengths and diameters it will be possible to tune the opening and closing of the valves to the frequency of the explosions. Further experimental investigation is needed to clarify the burning phase and to obtain more information with which the combustion tube sizes can be determined. Section 5 describes the proposed experiments more fully.

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Three of these 8 inch diameter engines with a tip speed of 600 feet per second are capable of supplying enough power to propel a 3600 pound gross weight helicopter including a 1200 pound payload (including pilot). The endurance of this machine would be about 3.0 hours at full load or about 6.2 hours with no payload. It is expected that helicopters propelled with the Multi-Jet will be able to compete quite favorably with the piston driven helicopter as well as making the overall construction much simpler and cheaper.

A full description of the jet unit and a comparison of various helicopter drive systems is given in References 3 and 4.

SECTION III

EXPERIMENTAL INVESTIGATION

3.1 Detonation

The detonation experiments were continued in the shock tube as described in the last progress letter (see Figure 22 in Reference 1).

Various ethylene-air mixtures were introduced into the expansion chamber and their combustability or ignitability was proved by their ignition by means of a spark. This spark was obtained by means of a small ignition system used on the small model airplane gas engines. The ethylene and air were introduced separately into the expansion chamber and were thoroughly mixed by means of a plunger which was passed through the tube a few times. When the mixture was ignited with the spark a very weak, muffled pop was heard and a very dull, blue flame, tinged with orange, was noticed issuing from the end of the tube. During these ignition tests, a single cellophane diaphragm was placed in its usual position in the shock tube and a single sheet was tied to the end of the expansion tube. The pressure obtained due to the ignition was barely able to blow off the piece of cellophane on the end of the tube, while the diaphragm was unbroken. After each ignition the tube was blown out by allowing compressed air to pressurize the compression chamber and rupture the single sheet of cellophane clamped in the diaphragm position.

The next step was to determine which of the above mixtures that proved to be ignitable by means of a spark could be detonated by means of a shock wave. The fuel-oxygen in the ethylene-air mixture was in the same range of ratios as those used by Shepherd in his experiments on ethylene-oxygen reported in Reference 7. These fuel-air mixtures could not be detonated by means of a shock wave. Pressures up to 100 psi were used in the compression chamber producing shock waves with a pressure ratio of 2.5. It was deduced then that the shock waves were not strong enough to detonate the ethylene-air mixtures. Since it was impossible to obtain higher pressures with the present air supply, stronger waves were not produced.

It was then decided that the experiments of Shepherd (Reference 7) should be repeated. He found that an 18% mixture of ethylene and oxygen could be detonated by means of a shock wave having a pressure ratio of 1.80 (produced in a shock tube having a compression chamber pressure of 65 psi.)

In our experiments the air in the expansion chamber was displaced by allowing the oxygen to flow through the tube for a few minutes. The tube was then closed and the ethylene was allowed to flow into the expansion chamber. The gases were then thoroughly mixed. Again shock waves up to a pressure ratio of 2.50 did not detonate the ethylene-oxygen mixtures. It was not clear why Shepherd's results could not be duplicated. To determine if the mixture was combustible, the spark was again used to ignite the mixture.

On ignition with the spark the mixture ignited and burned with explosive violence. It was deduced that detonation had occurred. The expansion chamber was 3½ feet long and 2 inches in diameter. A tube 4" in diameter surrounded the end of the expansion chamber to collect the exhaust gases and duct them out of the room. This 4" tube which was made of galvanized sheet iron was blown apart by the explosion. It was from this observation that it was deduced that detonation had occurred. A better method of detecting a detonation wave is needed.

The experiments in detonation have been discoutinued until more air pressure can be obtained to produce stronger shock waves.

3.2 Ceramic Burner Tests

Very simple burner tests have been set up to test the surface combustion in several different kinds of ceramic tubes. The purpose of these tests is to determine the actual conditions in a tube having a size which more closely approximates the tubes being used in the Multi-Jet.

The parameters being measured are air flow, fuel flow, inlet temperature, outlet temperature and tube temperature. Ethylene will be used in the first experiments. Minimum wall temperatures required to start the high mass flow burning will be investigated. Ignition limits of the fuel-air ratios will be determined.

Various ceramics will be tried such as Carbofrax[1], Stupalith[2], Metamic[3], and other materials that are available. The durability

---

1. A bonded silicon carbide refractory supplied by the Carborundum Co., Refractories Division, Perth Amboy, New Jersey

2. A lithium alumino-silicate composition supplied by Stupakoff Ceramic and Manufacturing Co., Latrobe, Pennsylvania

3. An aluminum oxide material with cobalt binder supplied by the Haynes Stellite Company, Division of Union Carbide and Carbon Corporation, 725 S. Lindsay St., Kokomo, Indiana

and the resistance to abrasion will be checked for these various materials and also their resistance to thermal shock.

The equipment for these tests is being assembled and the tests will be under way shortly.

BIBLIOGRAPHY

1. BITONDO, D. Preliminary Performance Analysis of the Pulse-Detonation-Jet Engine System Aerophysics Development Corporation, Pacific Palisades, California. Second Quarterly Progress Report No. ADC-102-2, July 1952

2. BITONDO, D. Preliminary Performance Analysis of the Pulse-Detonation-Jet Engine System Aerophysics Development Corporation, Pacific Palisades, California. First Quarterly Progress Report No. ADC-102-1, April (1952)

3. BITONDO, D. Summary Report on the Performance of a Multi-Jet Suitable for Helicopter Propulsion Aerophysics Development Corporation, Pacific Palisades, California. Technical Report No. ADC-102-4 October (1952)

4. BITONDO, D; KENDRICK, J. B. Preliminary Report on the Application of the Pulse-Det-Jet Engine to Helicopter Rotor Propulsion Aerophysics Development Corporation, Pacific Palisades, California Technical Report No. ADC 102-3 August (1952)

5. HICKS, JR., HENRY H.; WEIR, JR., ALEXANDER High Mass Flow Ceramic Ram-Jet Burner Aeronautical Research Center, Engineering Research Institute, University of Michigan Report No. UMM-73, December (1950)

6. HOWLAND, A. H.; SIMONDS, W.A. A Mechanism of Intense Gaseous Combustion The Gas Research Board, Beckenham, Kent, England Publication GRB 49 (1951)

7. SHEPHERD, W.C.F.; Ignition by Impulsive Pressures Third Symposium on Combustion, Page 301 Williams and Wilkins, Baltimore (1948)

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APPENDIX I

PERFORMANCE COMPUTATIONS FOR THE SUPERSONIC ENGINE

The computations of Tables I and II give the performance calculations for a 36" diameter Multi-Jet operating under an actual cycle. Most of the headings are self-explained. Table I gives the supersonic performance computations while Table II gives the subsonic performance computations. The numerical subscripts refer to the flow conditions during the different phases of the cycle. Referring to Table I and Figure 1, the subscript "o" refers to the ambient conditions of the flow; "1" refers to the flow conditions of the duct flow during scavenging; "2" refers to the flow conditions after the passage of the shock wave; "3" refers to the conditions in the closed combustion chamber after burning is completed and just before discharge. The primed symbol is used to denote the flow conditions of the same inlet phase after the flow has arrived at the rear of the tube and has been influenced by the frictional forces in the duct. It will be noted that $M_1 = 0.6$ for all cases and the tube diverges at a small angle in order to keep M_1 = constant throughout the length. Column 18 gives the average density of the fuel-air mixture in the tube. Column 19 gives the ideal total weight of air trapped in the tube. The weight of air per cycle must still be reduced by a correction factor that is determined from the opening and closing times of the valves. The total pressure obtained in the tube must be reduced by this correction factor. Tables I and II are similar up to column 19. In Table I (for the supersonic case) column 20 gives the expansion ratio of the high pressure gases during the discharge phase for the supersonic flight velocities $M_8 \geq 1.00$. The impulse during the discharge phase for the various expansion ratios of Column 20 for a given total pressure ratio $P_3+P_3'/2P_o$ is plotted in Figure 2. These curves are obtained from Figures 29 and 32 of Reference 2. The given flight Mach number determines the final pressure during discharge given by

$$\frac{P_1 + P_1'}{2(2.17)}$$

for the supersonic Mach numbers. Then in Figure 29, (Reference 2) P/P_7 would correspond to column 20. From Figure 29 (Reference 2) for a given maximum cycle pressure and the value of column 20, the value

$$\frac{\tau_{DISCH} A_N a_3}{VOL}$$

is obtained. For the given pressure ratio $\frac{P_7}{P_o} = \frac{P_3 + P_3'}{2 P_o}$

FIGURE 1

IDEAL CYCLE

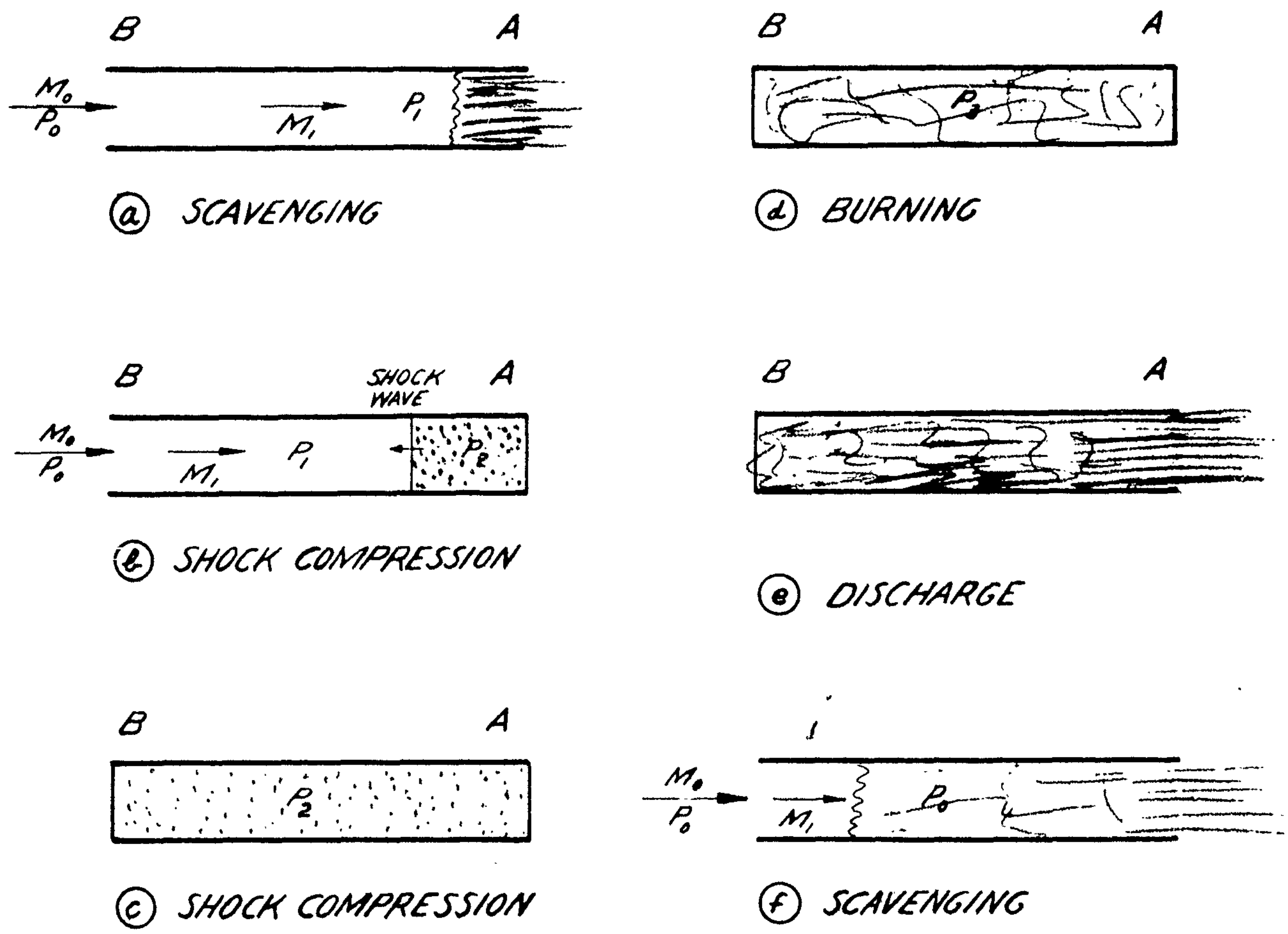

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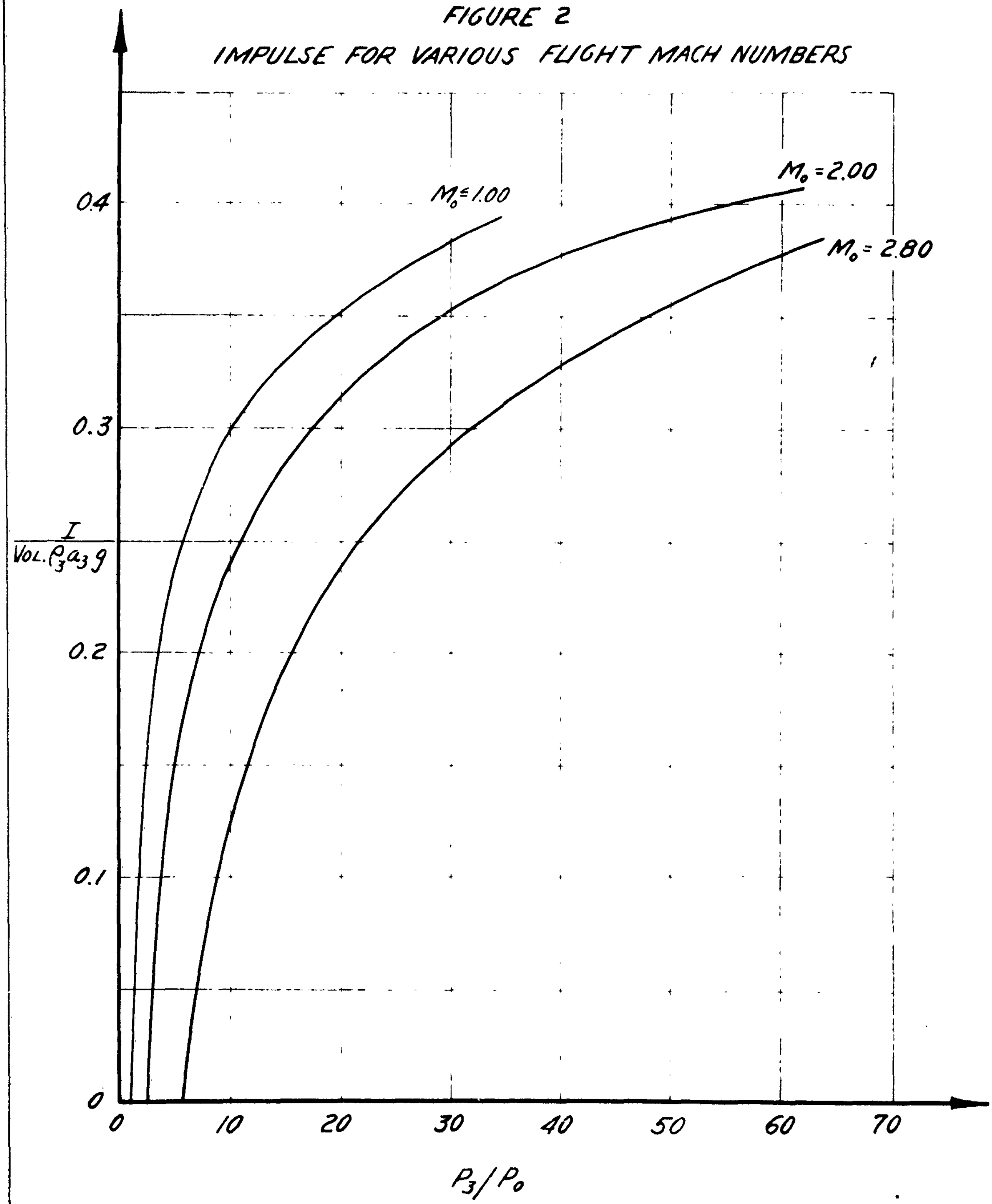
FIGURE 2

IMPULSE FOR VARIOUS FLIGHT MACH NUMBERS



on Figure 32 (Reference 2) the value of

$$\frac{\tau_{DISCH} \; A_N \, a_3}{Vol}$$

found on Figure 29 (Reference 2) is used to find the cut-off point on the thrust curve. The area is found under the thrust curve between the limits

$$0 \text{ to } \frac{\tau_{DISCH} \; A_N \, a_3}{Vol}$$

(given by Figure 29, Reference 2) and plotted on Figure 2 for the given value of M_o. In this way the curves for $M_o = 2.80$, 2.00 and 1.00 were obtained for Figure 2. From Figure 2 column 21 of Table I is obtained. By integration of the curve of Figure 30 (in Reference 2) from $\tau = 0$ to $\tau = \tau_{DISCH}$ (i.e. the value of τ found from Figure 29 (Reference 2) using P'/P_7 = column 20) gives the percentage of air discharged during the thrust phase (while the pressure dropped from P_3 to

$$\frac{P_1 + P_1'}{(P_3 + P_3')\, 2.17}$$

This can be seen from the following equation where w is the weight flow in lbs/sec.

$$\int \frac{w}{a_3 \rho_3 g A_N} \, d\, \frac{\tau_{DISCH} A_N a_3}{Vol} = \int \frac{w \, d\tau_{DISCH}}{g \rho_3 \, Vol}$$

where $\int w \, d\tau_{DISCH}$ is the weight of air discharged and ($g.\rho_3$.Vol) is the total weight of air contained in the tube before discharge. Column 23 gives the percentage of air discharged while 24 gives the weight of the remaining air. Since the gases are expanded isentropically in the tube during the discharge phase, the total temperature at the end of discharge is given in column 26. From the value of the total temperature in the tube the velocity of sound at a nozzle,(discharging at sonic velocity) can be found. This is the velocity of the remainder of the burnt gases during scavenging and is given by column 27. The value of τ_{SCAV} for column 28 is obtained from Figures 5 and 6 in Reference 1. The impulse of the scavenged gases as they are discharged is given in two parts. The gases first issue at a low nozzle pressure =

$$\frac{P_1 + P_1'}{2 \times 2.17}$$

After the pressure effects from the front valve reach the exhaust, the gases issue at a higher pressure = $P_1 + P_1'/2$. Approximately ¼ of the remainder of the gases leaves at the low pressure, the other 3/4 leaves at the higher pressure. It was assumed that the nozzle velocity remained the same during the whole scavenging period. Actually the pressure and temperature effects of the opening action of the inlet valve would increase the exit velocity from that given by column 27. This is neglected. Columns 29 and 30 give the two pressures. The impulse of these gases is

then computed in columns 31, 32 and 33 which when added give the impulse during scavenging (column 34). The impulse of the intake air is given by column 35. The net impulse is given by column 36. The total duration of the cycles are obtained from Figures 5 and 6 (Reference 1).

If the valves require a finite time "t" to open and close (Figure 3) then the mass that will flow into the tube while the inlet valve is open will be reduced from that of the ideal cycle. This also reduces the maximum cycle pressure. The reduction of these two parameters causes a reduction of thrust and specific thrust and an increase in specific fuel consumption.

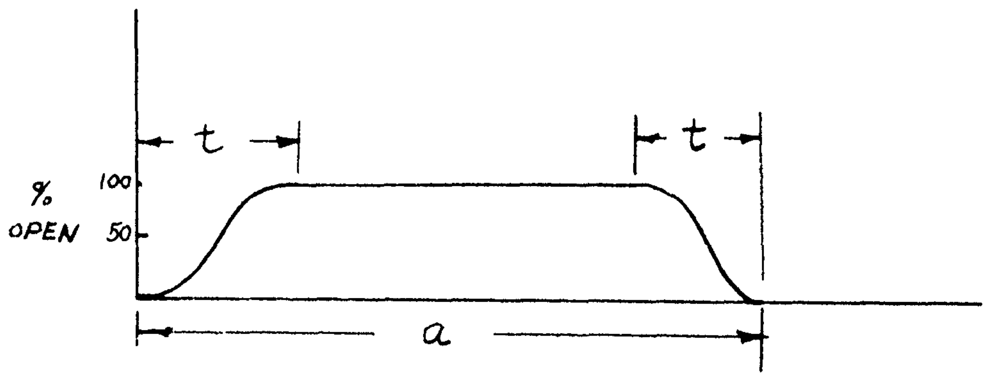


Figure 3

Valve Diagram

In Figure 3 "a" is the actual time the inlet valve is opened totally or partially. This is obtained from Figures 5 to 8. (Ref.1) It is assumed that if the inlet valve was fully open for the period of time "a" then the ideal mass would have entered the tube. Therefore, the ratio of the actual mass per cycle to the ideal mass per cycle would be

$$\frac{a-t}{a} = 1 - \frac{t}{a} \qquad (2)$$

The percentage decrease in thrust due to the reduction of mass flow is then

$$\frac{t}{a} \qquad (3)$$

The percentage decrease in maximum cycle pressure due to the decrease in mass per cycle is then t/a, and the further percentage decrease in thrust due to decrease in maximum cycle pressure is

$$K\frac{t}{a} \qquad (4)$$

Where K is given by Figure 26 of Reference 2.

The ideal thrust is then reduced by

$$\left(1-\frac{t}{a}\right)\left(1-K\frac{t}{a}\right) \tag{5}$$

The specific thrust is reduced by

$$\left(1-K\frac{t}{a}\right) \tag{6}$$

The specific fuel consumption is increased by

$$\left(1-K\frac{t}{a}\right) \tag{7}$$

The above considerations explain columns 38 to 46 and 49 and 50. The fuel-air ratio is found from Figure 10 of Reference 1.

Table II gives the computations for the subsonic flight velocities. Since columns 1 to 19 are similar to those of Table I they are not repeated.

Column 20 gives the expansion ratio during the discharge phase. Column 21 gives the impulse which is found from the curve of Figure 33 in Reference 2. For the subsonic flight Mach numbers, the pressure in the tube is allowed to drop to P_{ot} (where M_o defines P_{ot}) since the variation of P_{ot} for $0 \leq M_o \leq 0.75$ causes a very small variation in the impulse ($< 3\%$), differences in the impulse curve for the subsonic flight velocities were neglected. The curve of Figure 33 (Reference 2, was sufficiently accurate for this purpose. Columns 23, 24 and 25 of Table II are similar to columns 23, 24 and 25 of Table I. The exit nozzle pressure during scavenging was assumed to be equal to the velocity of the inlet fuel-air mixture (i.e. $M_1 a_1$). The impulse of the scavenged gases are therefore, given in column 26. The rest of the columns are similar to those of Table I.

Columns 51 to 57 of Table I and Columns 43 to 49 of Table II give the drag and thrust coefficients of a typical supersonic missile with a wing area of 425 square feet, powered by 4 of the 36" diameter engines. The thrust and drag coefficients are referred to the wing area and the drag coefficients are obtained from Figure 17(a) of Reference 1.

SUPERSONIC PERFORMANCE OF THE PULSE-DETONATION-JET

TABLE I

COLUMN	1	2	3	4	5	6	7	8	9	10
FUNCTION	M_o	T_3 °R	P_o psf	P_{ot} psf	P_{it} psf	P_1 psf	P_1' psf	P_2 psf	P_2' psf	T_{ot} °R
OPERATION							⑥ × 0.97	⑥ × 2.17	⑧ × 0.97	
	2.80	3960	2116	57,300	37,200	28,700	27,840	62,900	61,050	1330
		2960								
		1960								
	2.00	3960	2116	16,540	14,900	11,500	11,150	25,700	24,400	935
		2960								
		1960								
	1.00	3960		4,010	4,010	3,100	3,003	6,800	6,580	623
		2960								
		1960								

TABLE I (Cont'd)

1	2	11	12	13	14	15	16	17	18	19
M_o	T_3 °R	T_1 °R	T_2 °R	T_3/T_2	P_3 PSF	P_3' PSF	P_3 AVG	$\frac{P_3\ AVG}{P_o}$	$\gamma\ \rho_{3\,AVG}$	W (18)×VOL
			(11) × 1.27		(8) × (13)	(9) × (13)	$\frac{(14) \times (15)}{2}$	$\frac{(16)}{3}$	lbs/ft³	lbs/CYCLE (18) × .1211
2.80	3960	1240	1575	2.51	158,000	153,300	155,500	73.8	.738	.0896
	2960			1.878	118,200	114,800	116,500	55.2	.738	.0896
	1960			1.245	78,300	76,100	77,200	36.6	.738	.0896
2.00	3960	872	1108	3.68	92,700	89,800	91,250	43.2	.433	.0525
	2960			2.67	68,700	65,200	66,950	31.7		
	1960			1.77	45,500	43,200	44,350	21.0		
1.00	3960	581	738.5	5.36	36,470	35,280	35,875	17.0	.1705	.02065
	2960			4.01	27,270	26,380	26,825	12.7		
	1960			2.65	18,050	17,450	17,750	8.41		

VOL. OF 1 ROW OF TUBES FOR A 34" DIAM ENGINE = 0.1211 cu. ft.

1 ENGINE CONTAINS 32 ROWS OF TUBES

TABLE I (Cont'd)

1	2	20	21	22	23	24	25	26	27	28
M_o	T_3 °R	$\frac{P_1+P_1'}{2P_{3AVG}}\times 2.17$	$\frac{I_v}{a_3\, g\, \rho_3\, Vol}$	I_v lb. SECS	W DISCHARGED	W REMAINING	W DISCHARGED DURING SCAV.	T_t END OF DISCHARGE	VEL. OF EXHAUST FPS	τ_{SCAV}
				(21)×(19)×a_3	%	%	(24)×(19) lbs	$(20)^{2/7}\times T_3$	$46.95\times\sqrt{.875\times(26)}$	
2.80	3960	.084	.0403	10.60	.8165	.1835	.01642	1950	1930	.00190
	2960	.112	.0368	8.45	.7848	.2152	.0193	1580	1740	
	1960	.169	.0318	5.90	.718	.282	.0252	1180	1508	
2.00	3960	.0522	.0382	5.88	.646	.154	.0081	1700	1800	.00212
	2960	.078	.03585	4.82	.8264	.1736	.00912	1430	1655	
	1960	.118	.03210	3.49	.776	.224	.0118	1060	1440	
1.00	3960	.0388	.0344	4.085	.878	.122	.00252	1560	1730	.00250
	2960	.0524	.0301	1.60	.846	.154	.00318	1270	1560	
	1960	.0792	.0268	1.15	.826	.174	.00360	950	1340	

TABLE I (Cont'd)

1	2	29	30	31	32	33	34	35	36	37
M_0	T_3 °R	SCAV. PRESS. 1st 1/4 OF PHASE PSF	SCAV. PRESS LAST 3/4 OF PHASE	PRESS. IMP. 1ST 1/4 OF SCAV.	PRESS. IMP. LAST 3/4 OF SCAV.	VEL. IMP. DURING SCAV.	I_s lb SECS	INTAKE IMPULSE	I_{NET} lb SECS	τ_{TOT}
		$\frac{(6)+(7)}{2 \times 2.17}$	$\frac{(6)+(7)}{2}$ PSF	$((29)-2116)\,0.0958 \times \frac{t_{SCAV}}{4}$ lb SECS	$((30)-2116)\,0.0958 \times \frac{3\tau_{SCAV}}{4}$	$\frac{(25)\times(27)}{32.2}$	$(31)+(32)+(33)$	$\frac{(19)\times M_0 \times a_0}{32.2}$	$(22)+(34)-(35)$	SECS
2.80	3960	13,000	28,270	0.517	3.73	0.985	5.232	8.70	7.132	.00589
	2960					1.04	5.287		5.037	
	1960					1.18	5.427		2.627	
2.00	3960	5,210	11,325	0.164	1.48	0.458	2.102	3.64	4.549	.00681
	2960					0.468	2.112		3.092	
	1960					0.528	2.172		2.022	
1.00	3960	1,530	3,051	- .0366	.175	.135	.270	0.716	1.639	.00806
	2960					.154	.289		1.17	
	1960					.150	.285		.716	

AREA OF 1 ROW OF TUBES OF A 34" DIA. ENGINE = .0958 ft^2

TABLE I (Cont'd)

1	2	38	39	40	41	42	43	44	45	46	47
M_o	T_3 °R	"a"	"t"	t/a	K	$1-\frac{t}{a}$	$1-K\frac{t}{a}$	$(1-\frac{t}{a})(1-K\frac{t}{a})$	THRUST OF 36" D. ENGINE	SPECIFIC THRUST lbs/lb AIR/SEC	T_3-T_2
		SECS	SECS						$\frac{36 \times 44 \times 32}{37 \times 0.893}$	$\frac{36}{19} \times 43$	°F
2.80	3960	.00285	.00050	.176	.015	.824	.997	.822	36,650	79.5	2385
	2960				.025		.996	.821	25,150	56.0	1385
	1960				.036		.994	.820	13,125	28.7	385
2.00	3960	.00326	.00055	.168	.031	.832	.995	.829	19,800	85.5	2852
	2960				.040		.993	.827	13,400	57.8	1852
	1960				.055		.991	.825	8,790	36.0	852
1.00	3960	.00395	.000680	.172	.065	.828	.99	.82	5,980	78.5	3222
	2960				.105		.98	.815	4,350	55.0	2222
	1960				.230		.96	.805	2,570	33.2	1222

$$\frac{\text{AREA OF 34" DIAM ENGINE}}{\text{AREA OF 36" DIAM ENGINE}} = 0.893$$

TABLE I (Cont'd)

1	2	48	49	50	51	52	53	54	55	56	57
M_o	T_3 °R	FUEL / AIR	SPECIFIC FUEL CONSUMPTION lbs/HR/lb THRUST	THRUST / FRONTAL AREA	C_T - 4 ENGINES BASED ON 425	C_D SEA LEVEL	C_D 5000' ALT.	C_D 15000' ALT	C_D 25000' ALT	C_D 35000' ALT	DRAG SEA LEVEL
		lb/lb	$\frac{3600 \times 48}{46}$	$\frac{lbs}{ft^2}$	$\frac{4 \times 45}{q \times 425}$						lbs.
2.80	3960	.0338	1.53	5,050	.0297	.0230	.0231	.0232	.0235	.0241	
	2960	.0170	1.09	3,560	.0204						
	1960	.0044	0.552	1,820	.0107						
2.00	3960	.0440	1.84	2,790	.0314	.0258	.0259	.0260	.0270	.0290	
	2960	.0228	1.42	1,900	.0213						
	1960	.0099	0.990	1,175	.0139						
1.00	3960	.0455	2.09	846	.0380	.0302	.0327	.0358	.0435	.0636	19,000
	2960	.0276	1.80	606	.0276						
	1960	.0133	1.44	364	.0163						

SUBSONIC PERFORMANCE OF THE PULSE-DETONATION-JET

TABLE II

COLUMN	1	2	3	4	5	6	7	8	9	10	11
FUNCTION	M_0	T_3 °R	P_0 psf	P_{0t} psf	P_{1t} psf	P_1 psf	P_1' psf	P_2 psf	P_2' psf	T_{0t} °R	T_1 °R
OPERATION							6 × 0.97	6 × 2.17	8 × 0.97		
	0.75	3960	2116	3070	3070	2370	2300	5200	5050	577	539
		2960									
		1960									
	0.50	3960		2510	2510	1940	1880	4250	4120	545	509
		2960									
		1960									
	0.25	3960		2210	2210	1710	1660	3750	3640	525	490
		2960									
		1960									
	0	3960		2116	2116	1635	1587	3580	3477	519	484

TABLE II (Cont'd)

1	2	12	13	14	15	16	17	18	19	20	21
M_o	T_3 °R	T_2 °R	T_3/T_2	P_3 psf	P_3' psf	$P_{3\,AVG}$	$\frac{P_{3\,AVG}}{P_o}$	$g\rho_{3\,AVG}$	W = (18) x Vol	$\frac{P_o t}{P_{3\,AVG}}$	$\frac{I_v}{a_3\, g\rho_3\, Vol}$
		(11) x 1.27		(8) x (13)	(9) x (13)	$\frac{(14)+(15)}{2}$	(16)/(3)	lbs/ft³	lbs/cycle (18) x 0.1211		
0.75	3960	684	5.79	30,100	29,200	29,650	14.05	.149	.01709	.1035	.0330
	2960		4.33	22,480	21,820	22,150	10.5			.139	.0300
	1960		2.87	14,910	14,470	14,690	6.95			.209	.0258
0.50	3960	646	6.03	25,600	24,800	25,200	11.9	.120	.01453	.100	.0317
	2960		4.58	19,440	18,840	19,140	9.05			.131	.0291
	1960		3.04	12,910	12,510	12,710	6.03			.1975	.0236
0.25	3960	622	6.36	23,900	23,200	23,550	11.13	.1119	.01357	.094	.0310
	2960		4.76	17,850	17,320	17,585	8.32			.126	.0281
	1960		3.16	11,830	11,490	11,660	5.53			.190	.0224
0	3960	615	6.44	23,060	22,390	22,725	10.78	.1078	.01305	.093	.0308
	2960		4.81	17,250	16,750	17,000	8.05			.124	.0278
	1960		3.19	11,430	11,090	11,260	5.33			.188	.0217

VOL. OF 1 ROW OF TUBES FOR 34" DIAM. ENGINE = 0.1211 ft³

1 ENGINE CONTAINS 1 ROW OF TUBES

TABLE II (Cont'd)

1	2	22	23	24	25	26	27	28	29	30	31
M_o	T_3 °R	I_{V} lb sec	W DISCHARGED	W REMAINING	W DISCHARGED DURING SCAV.	I_S	INTAKE IMPULSE lb SECS	I_{NET} lb SECS	τ_{TOT}	"a"	"t"
		(21)×(19)×a_3	%	%	(24)×(19) lbs	$\frac{(25)\times M_1 \times a_1}{32.2}$	$\frac{(19)\times M_o \times a_o}{32.2}$	(22)+(26) −(27)	SECS	SECS	SECS
0.75	3960	1.65	.798	.202	.00345	.0733	0.440	1.283	.00835	.00405	.000710
	2960	1.31	.750	.25	.00427	.0905		.96			
	1960	0.915	.68	.32	.00547	.116		.591			
0.50	3960	1.35	.799	.201	.00292	.060	0.252	1.158	.00880	.00413	.000750
	2960	1.08	.758	.242	.00352	.0722		.900			
	1960	0.71	.689	.311	.00452	.0927		.551			
0.25	3960	1.23	.806	.194	.00262	.0533	0.105	1.178	.00890	.00425	.000760
	2960	0.976	.78	.22	.00298	.0606		.931			
	1960	0.629	.733	.267	.00362	.0736		.597			
0	3960	1.18	.83	.17	.00222	.0442	0	1.224	.00900	.00435	.000765
	2960	0.929	.799	.201	.00262	.0521		.981			
	1960	0.586	.750	.25	.00326	.0650		.651			

TABLE II (Cont'd)

1	2	32	33	34	35	36	37	38	39	40	41
M_o	T_3 °R	t/a	K	$1 - t/a$	$1 - K\frac{t}{a}$	$(1-\frac{t}{a})(1-K\frac{t}{a})$	THRUST OF 36" D. ENGINE	SPECIFIC THRUST lbs/lb AIR/SEC	$T_3 - T_2$	FUEL / AIR	SPECIFIC FUEL CONSUMPTION lbs/HR/lb THRUST
							$\frac{28 \times 36 \times 32}{29 \times 0.893}$	$\frac{28}{19} \times 35$	°F	LB. / LB.	$\frac{3600 \times 40}{38}$
0.75	3960	.175	.090	.825	.984	.821	4550	74.0	3276	.0462	2.14
	2960		.160		.977	.815	3360	54.8	2276	.0279	1.83
	1960		.370		.935	.781	1990	32.4	1276	.0138	1.53
0.50	3960	.182	.115	.818	.979	.80	3770	77.9	3314	.0468	2.16
	2960		.20		.964	.789	2900	59.7	2314	.0283	1.71
	1960		.53		.90	.735	1650	34.2	1314	.0142	1.50
0.25	3960	.179	.13	.821	.977	.801	3820	84.7	3338	.0470	2.00
	2960		.24		.957	.785	2950	65.7	2338	.0285	1.56
	1960		.675		.88	.723	1760	38.8	1338	.0144	1.34
0	3960	.176	.14	.824	.975	.803	3930	91.5	3345	.0470	1.85
	2960		.26		.954	.785	3070	71.6	2345	.0785	1.43
	1960		.75		.87	.716	1850	43.5	1345	.0144	1.19

TABLE II (Cont'd)

1	2	42	43	44	45	46	47	48	49
M_o	T_3 °R	THRUST / FRONTAL AREA	C_T - 4 ENGINES BASED ON 425 ft^2	C_D SEA LEVEL	C_D 5000' ALT	C_D 15000' ALT	C_D 25000' ALT	C_D 35000' ALT	DRAG SEA LEVEL
		$\frac{lbs}{ft^2}$	$\frac{4 \times 37}{\gamma \times 425}$						lbs
0.75	3960	642	.0515	.0156	.0242	.0351	.0615	.131	5,530
	2960	476	.0380						
	1960	281	.0225						
0.50	3960	534	.0960	.0500	.0651	.1226	.2646	.2646	7,850
	2960	410	.0738						
	1960	234	.0420						
0.25	3960	540	.388	.582	.825	1.7556	4.09	10.12	22,900
	2960	417	.300						
	1960	249	.179						
0	3960	555							
	2960	435							
	1960	262							